A BRIEF SURVEY OF ATTITUDE CONTROL SYSTEMS FOR SMALL SATELLITES USING MOMENTUM CONCEPTS

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High-accuracy pointing capabilities are desired for many three-axis stabilized small satellites. Momentum-based attitude control system actuators, initially developed for larger satellites, are being utilized by small satellites to meet these pointing requirements. This paper provides an overview of momentum devices available for small satellite applications and three-axis attitude control system (ACS) configurations using these devices. Factors affecting the selection and sizing of ACS components are also addressed. Included are suggestions for potential ACS improvements and cost-saving measures that will make momentum devices more accessible to the small satellite community.

INTRODUCTION

Small satellites are currently being used in various mission/payload regimes that require high-accuracy pointing capabilities. For example, small three-axis stabilized satellites are used to accurately pinpoint their instruments to specific objects or regions of space. Also, small communication satellites, especially those employing multiple narrow-beam antennas, require tight pointing accuracies to ensure adequate antenna gain. Attitude control system technologies developed for larger satellites are being utilized to meet increasingly stringent pointing requirements. At present, pointing accuracies of 1 degree or less are quite common for both spin and three-axis stabilized small satellites.

The selection of an attitude control system (ACS) is a function of many factors, including mission objectives, orbit, and available system budgets. Stabilization systems using momentum and reaction wheels as control torque sources are well-suited to small satellite applications due to their proven performance, relative simplicity, versatility, and capability of providing high-accuracy pointing control. This paper addresses some of the factors to be considered when selecting an ACS and focuses on the utilization of various momentum-based actuators to meet typical ACS requirements for small satellites. Topics covered include a discussion about momentum and torque requirements, a complete overview of available momentum devices, and examples of ACS that use these devices. An example of wheel selection and sizing is included, along with a section on ACS improvements and cost reductions for small satellites.

When momentum-based systems are used to stabilize and control the attitude of a satellite, an auxiliary torquing system is included to desaturate the wheels. The auxiliary torquing system

can employ thrusters, gravity gradient, solar pressure, or magnetic torquers. This paper presents an overview and comparison of various desaturation schemes.

FACTORS AFFECTING ACS CONFIGURATION

The selection of an ACS configuration, and the sizing of its components, is a complex function of various parameters that are typically at odds with one and other. Some of these parameters are constrained, while others are variable within a certain range. Fig. 1 shows factors in the selection of an ACS.
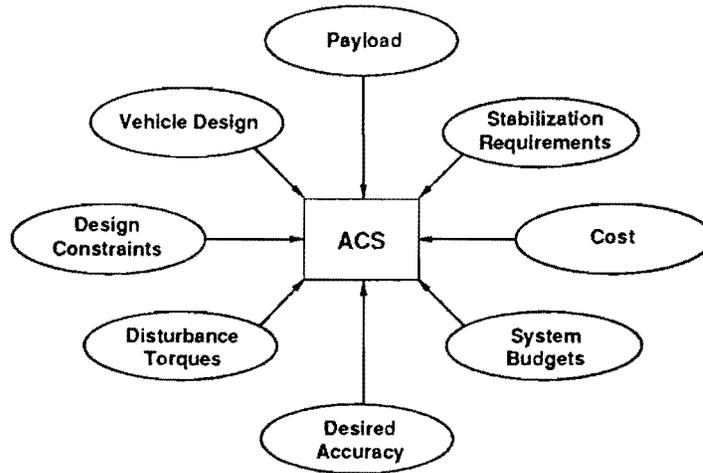
Fig. 1 Factors Affecting ACS Selection

Payload Stabilization and Pointing Requirements

Mission objectives, tempered by available budgets, determine payload instrumentation, orbit selection, stabilization, and pointing requirements. These parameters set the requirements and the range of variability for the ACS and its components.

Mission and system requirements dictate stabilization and place the ACS into either a spin or three-axis stabilization configuration. Spin stabilization is generally less complex and less expensive than three-axis stabilization and is well-suited for some scientific experiments. Three-axis stabilization, however, is required for some communications, reconnaissance, Earth observation, and astronomy missions.

The Ball Scout Satellite (BSS)[1] is a small satellite designed to accommodate both Earth-oriented and astro science missions. The mission design set for the BSS, which is representative of those for small satellites in general, requires both spin and three-axis stabilization capabilities. A summary of the BSS mission design set is listed below. All ACS configurations for the various missions have requirements for pointing control capability of 0.2 deg and attitude control knowledge of 0.1 deg.

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- Earth-Oriented Missions

  - Nadir pointer (polar sun-synchronous orbits)
  - Nadir pointer (polar nonsynchronous orbits)
  - Orbit normal spinner (polar sun-synchronous orbits)

- Astro-Oriented Missions

  - Zenith pointer (polar sun-synchronous orbit)
  - Zenith pointer (polar nonsynchronous orbit)
  - Celestial pointer (polar nonsynchronous orbit) with slow or fast retargetting capability
  - Sun-line spinner (polar sun-synchronous orbit)
  - Sun-line spinner (equatorial nonsynchronous orbit)

All orbits are 400 km circular.

Vehicle Design

The mass moments of inertia of a space vehicle are critical to the selection and operation of an attitude control system. Moments of inertia affect the sizing of the actuators and determine the effect of disturbance torques on pointing and stability. Simulations of the equations of motion show the relationship between vehicle inertias, disturbance torques, and pointing requirements. Inertias may be fixed by design constraints or may be a design option that permits optimization for a given orientation and/or disturbance torque profile.

Other vehicle design parameters influence the effects of the expected environmental torques. For example, surface properties of the vehicle determine the effects of solar radiation pressure. The location of the center of pressure with respect to the center of mass, as well as the area, and the coefficient of drag determine the effects of aerodynamic drag torques associated with low-altitude orbits. Environmental torques are not necessarily detrimental since they can be used for passive stabilization through proper vehicle design.

The configuration of the vehicle may impose design constraints on the ACS in the form of system budgets for weight, power, and volume. These constraints are often imposed on the vehicle by available launch opportunities, especially for small satellites.

System Budgets

System budgets for power, weight, and volume and their allotment for attitude control define an envelope for workable systems. A number of configurations may fall within the envelope, thus giving various options to the small satellite designer. Identifying workable configurations and determining compliance with budget restrictions at the outset of the system selection enables the system engineer to identify design margins and potential shortfalls.

A brief analysis of recent small satellites gives an idea of the magnitude of small satellite budgets for power, weight, and volume. The following data shows pertinent budget information, as well as available ACS information, for various small satellite buses.

AMSAT (Amateur Satellite)[2]

Summaries include all Amateur Satellite programs, OSCAR I through Phase IVA.

Bus mass (kg):	4.5 to 125.0
Bus power rating (watts):	0.14 to 43.0

AMSAT ACS Configurations	System Mass (kg)	System Power (W)
None	4.5 - 16.0	0.10 - 1.00
Spin	16.0	3.0
Spin, passive magnets, and lossy dampers	18 - 50	0.15 - 11.0
Spin- and computer-operated magnets	90 - 125	41.0 - 43.0
Gravity gradient boom	60	25.0
Gravity gradient boom and active magnets	60	25.0
Body stabilized; reaction control systems	400	230.0

Note that some of the AMSAT satellites have one or more transponders, while others have none. Therefore, relative changes in bus mass and power are not exclusively related to changes in the ACS configuration.

Applied Physics Laboratory (APL) Spacecraft[3]

SAS (Small Astronomy Satellite)

Satellite	Mass (kg)	ACS Configuration
SAS-1	185	Single momentum wheel
SAS-2	185	Dual spin with magnetic coils for spin and precession
SAS-3	189	Same as above

Power Summary for recent APL spacecraft:

Minimum orbit average power: 34 - 282 W
Power generated per unit weight of solar panels: 2.78 - 10.83 W/kg

Small Explorer Satellite[4]

Desired performance characteristics for the satellite bus.

Mass: 100 kg
Volume: cylinder with diameter = 75 cm, length = 45 cm.
Pointing: 0.5 deg (yaw); 0.01 deg (pitch and roll)

ACS configuration: momentum stabilization using two orthogonal momentum wheels and various sensor suites, depending on payload

Ball Scout Satellite (BSS)[2]

Pointers (nadir, zenith, and celestial):

Bus mass (kg):	112.5 - 136.5		Bus power (W):	43.5 - 48.3	
ACS mass (kg):	6.4 - 12.8		ACS power (W):	5.6 - 7.7	
Percent of bus mass:	5.7 - 10.9		Percent of bus power:	12.2 - 15.9	

Spinners (orbit normal and sun-line):

Bus mass (kg):	115.4 - 133.2		Bus power (W):	43.5 - 49.5	
ACS mass (kg):	7.1 - 7.4		ACS power (W):	6.3 - 8.6	
Percent of bus mass:	5.3 - 6.4		Percent of bus power:	13.5 - 17.4	

Orion[5]

Bus mass: 170 lb (270 lb total)
Bus power: 50 W
Volume: Cylinder of diameter = 19 in., length = 35 in. (volume restricted by
            the get-away special (GAS) cannister envelope)

ACS: Active nutation control using 0.1 lb thrusters (angles bounded by
            0.5 deg and 3.0 deg)
            Consumes approximately 0.5 lb/day

Orbit and Environmental Disturbance Torques

The orbit of a small satellite, including elevation, inclination, and eccentricity, and the satellite configuration, determines the type of environmental torques to be expected during satellite operation. Environmental torques, as mentioned previously, can be modeled to determine both passive stabilization feasibility and disturbance torque characteristics. Environmental torque magnitudes are one of several factors that affect the sizing of ACS components.

Shrivastava and Modi[6] outline sources of environmental forces and torques. A summary of environmental torque disturbance models from this reference and pertinent comments are listed below.

Gravity Gradient Torque

Equation: $\underline{T}g = 3 [(I_3 - I_2) \, mn \, \underline{i} + (I_1 - I_3) \, \ell n \, \underline{j} + (I_2 - I_1) \, \ell m \, \underline{k}] \, \dot{\theta}^2/(1 + e \cos \theta)$ (1)

Assumptions: Higher-order terms neglected.
              Rigid spacecraft assumed.

Variables: $\underline{T}g$ = gravity gradient torque

            θ = true anomaly

e = orbital eccentricity

I_1, I_2, I_3 = principal moments of inertia

ℓ, m, n = direction cosines of the Earth satellite line with respect to the principal body axes

Comments:

- Gravity gradient torque vanishes if:

  - Any two direction cosines are zero.
  - $I_1 = I_2 = I_3$
  - Any two inertias are equal and one direction cosine is zero.

- Gravity gradient torque is a disturbance for a three-axis stabilized satellite unless the configuration is such that its maximum moment-of-inertia axis is perpendicular to the orbit, and the minimum moment-of-inertia axis is along the local vertical.

- Gravity gradient can be used as a passive form of stabilization, as well as a momentum desaturation technique. For desaturation, the concept is attractive due to its simplicity. However, for stabilization, the disadvantages of this concept are that it is highly susceptible to other environmental disturbances and must include a low-accuracy design of an effective damper.

- The eccentricity of an elliptical orbit acts as a periodic forcing term that makes the disturbing effect of a gravity gradient more pronounced.

Aerodynamic Drag Torque

Equation: $\underline{T}_A = \underline{\varepsilon} \times \underline{F}_A$ (2)

$$\underline{F}_A = 0.5\, \rho_a\, V^2\, A\, C_D$$

Assumptions: Equation for $\underline{F}_A$ is an approximation.

Variables: ε = position vector of the center of pressure with respect to the center of mass

ρ_a = atmospheric density

V = velocity of the satellite relative to the atmosphere

A = characteristic area

C_D = coefficient of drag

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Comments:

- Aerodynamic forces are significant for orbital altitudes up to 800 km.

- Aerodynamic torque is generally the dominant environmental disturbances for low earth orbits.

- Difficulties in modeling the atmosphere are encountered due to large variations in density with height, relative position of the sun, and solar activity. Also, atmospheric rotation and the oblateness of the Earth must be accounted for if reasonable accuracy is desired.

Solar Radiation Torque

Equation: $\underline{F}_s = (S/C) \mid \cos \xi \mid \{(1 - \tau - \rho) \underline{n} + 2\rho \cos \xi \underline{p}\} \, dA$ (3)

Assumptions: Solar radiation torque calculation requires knowledge of exact satellite geometry.

Variables: $\cos \xi = \underline{n} \cdot \underline{p}$

     $\underline{n}$ = unit vector normal to the surface

     $\underline{p}$ = direction of the incident ray

     S = solar constant

     C = speed of light

     τ, ρ = transmissibility and reflectivity of the surface, respectively

     dA = differential area experiencing solar radiation

Comments:

- Solar radiation torque can be the dominant source of disturbance torque for geostationary spacecraft.

- The model for force due to solar pressure may be complicated by the Earth's shadow, secondary reflection, and the degradation of vehicle surfaces.

Magnetic Torque

Equation: $\underline{T}_M = \underline{M} \times \underline{B}$ (4)

Assumptions: none

Variables: $\underline{M}$ = dipole moment of the satellite (determined experimentally)

$\underline{B}$ = the local geomagnetic induction

Comments:

- Interaction with the Earth's magnetic field is a major source of disturbance for satellites below 1500 km.

- A simplified model of the Earth's magnetic field is a small magnetic dipole with a strength of 8.06 x 10^5 Wb-m, located at the center of the Earth, whose axis is tilted at 11.5 deg with respect to the polar axis.

- Magnetization and demagnetization of on-board ferromagnetic material leads to hysteretic damping.

- Eddy currents generated in a spin-stabilized satellite result in torques about the spin axis.

- It is possible to reduce the undesirable influence of magnetic torque by minimizing:

    - use of ferromagnetic materials
    - the satellite's dipole moment through correct design of electrical circuits
    - eddy currents by reducing thickness and interrupting continuity of conducting structural material

The environmental torque models presented here can be used in dynamical analyses, design of the ACS, orbital simulation, performance analyses, and attitude determination.

MOMENTUM AND TORQUE REQUIREMENTS

Three main contributors to system momentum and torque requirements are:

- environmental torques

- vehicle maneuvers

- reaction torques

Environmental disturbance torques discussed in the previous section (including gravity gradient and aerodynamic, magnetic, and solar pressure) fluctuate in a cyclic manner over the orbital period. The effects of these torques on the vehicle can be nulled through momentum exchange with a momentum device. Momentum requirements vary sinusoidally, as illustrated in Fig. 2, and have an amplitude corresponding to the maximum amplitude of the expected disturbance. The torque disturbance models can be used to determine the types and magnitudes of disturbance torques that can be expected for a given orbit.
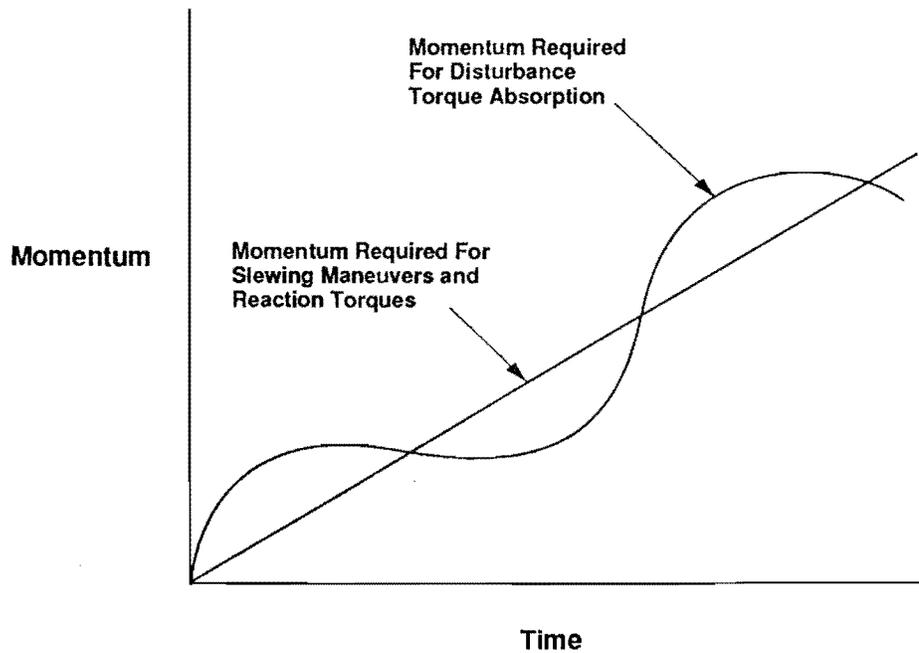
Fig. 2 Typical Vehicle Momentum Requirements

Additional momentum is required if vehicle slew maneuvers are desired. The amount of momentum and torque required for slew is determined by:

$$\underline{H} = \underline{I}_v(\omega)_v \quad \text{and} \quad \underline{T} = \underline{I}_v(\alpha)_v \tag{5}$$

where $\underline{T}$ = torque

$\underline{H}$ = momentum

$\underline{I}_v$ = vehicle inertia

ω_v = vehicle angular rate

α_v = vehicle angular acceleration

The required slewing time depends on the wheel capacity, the current momentum bias, and any attitude or attitude rate limits which may be imposed. Fig. 3 shows a diagram of torque, wheel momentum, body rate, and angular position for an idealized system.

Reaction torques acting on the vehicle due to antennas, solar arrays, scanning instruments, pumps, data recorders, etc must also be considered when selecting and sizing a wheel. Frequency and magnitudes of reaction torques must be determined to adequately size the momentum system.

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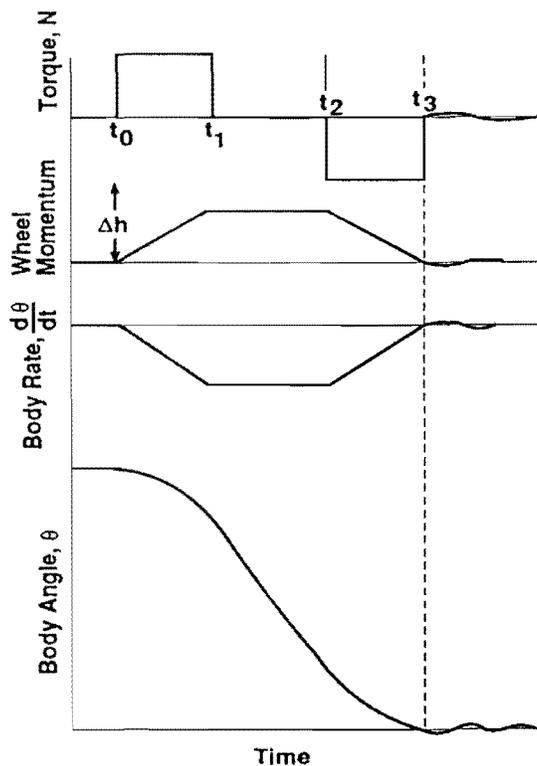
Fig. 3 Example of a Single-Axis Slew Maneuver

Momentum and torque requirements associated with reaction torques can be determined by using the following:

$$\underline{H} = \underline{I}_a(\omega)_a \quad \text{and} \quad \underline{T} = \underline{I}_a(\alpha)_a \tag{6}$$

where $\underline{I}_a$ = the inertia of the active component

ω_a = the angular rate of the active component

α_a = the angular acceleration of the active component

An orbital analysis of momentum requirements will yield the maximum amount of momentum required to handle the expected torques. In this analysis, the momentum required for cyclic disturbances is added to the cumulative momentum requirements associated with slew maneuvers and reaction torques (Fig. 2).

MOMENTUM-BASED ATTITUDE CONTROL ACTUATORS

Momentum devices can be used in various capacities in an attitude control system. Typical applications include storing and transferring angular momentum during vehicle slew maneuvers, absorbing disturbance torques, and stabilization of the spacecraft. Unfortunately, the terminology associated with momentum devices is not uniform. The following terminology has been adopted for this paper.[7]

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- **Momentum wheel**: momentum device designed to operate at a biased or nonzero momentum. It provides a variable momentum storage capability about its rotation axis, which is usually fixed in the vehicle, and gyroscopic stability about the transverse axes.

- **Reaction wheel**: momentum device with a vehicle-fixed axis, designed to operate at zero bias.

- **Control moment gyro (CMG)**: single- or double-gimballed wheel spinning at a constant rate. The gimbals allow control of the direction of the momentum vector in the spacecraft body.

- **Gimballed momentum wheel**: double-gimbal CMG with a variable-speed wheel.

- **Scanwheel**: (Trademark of ITHACO, Inc.) A horizon scanner/momentum-wheel combination.

Reaction and momentum wheels typically consist of a rotor, bearing unit, electric drive motor, and associated electronics. CMGs require, in addition to these components, a gimbal system and gimbal drive electronics. Wheel-speed controllers operate in one of two modes: constant wheel speed or torque control. In the constant wheel speed control mode, the wheel's angular rate is automatically maintained at a desired value. In the torque control mode, wheel speed is modulated through closed loop control in response to an error signal.

The performance of a wheel system depends on the angular momentum of a spinning wheel:

$$\underline{h} = \underline{I} \cdot \underline{\omega} \tag{7}$$

where $\underline{I}$ is the mass moment of inertia, and $\underline{\omega}$ is the angular velocity of the wheel. (The attitude coordinate system is shown in Fig. 4). The torque, $\underline{T}$, associated with a wheel system is equal to the rate of change of the angular momentum with respect to time

$$\underline{T} = \frac{d\underline{h}}{dt} \tag{8}$$

Since CMGs operate at a constant spin rate, momentum exchange is a result of a change in the direction of the momentum vector due to gimaballing. CMG output torque is equal to $\underline{\Omega} \times \underline{h}$, where $\underline{\Omega}$ is the gimbal angular rate. Output torque from momentum devices may have components in yaw, roll, and pitch directions, depending on the location of the momentum vector in the satellite.

Various types and combinations of momentum devices can be used to meet pointing and stabilization requirements for small satellites. Before specific configurations are discussed, a brief description of the operational characteristics of each device is given.
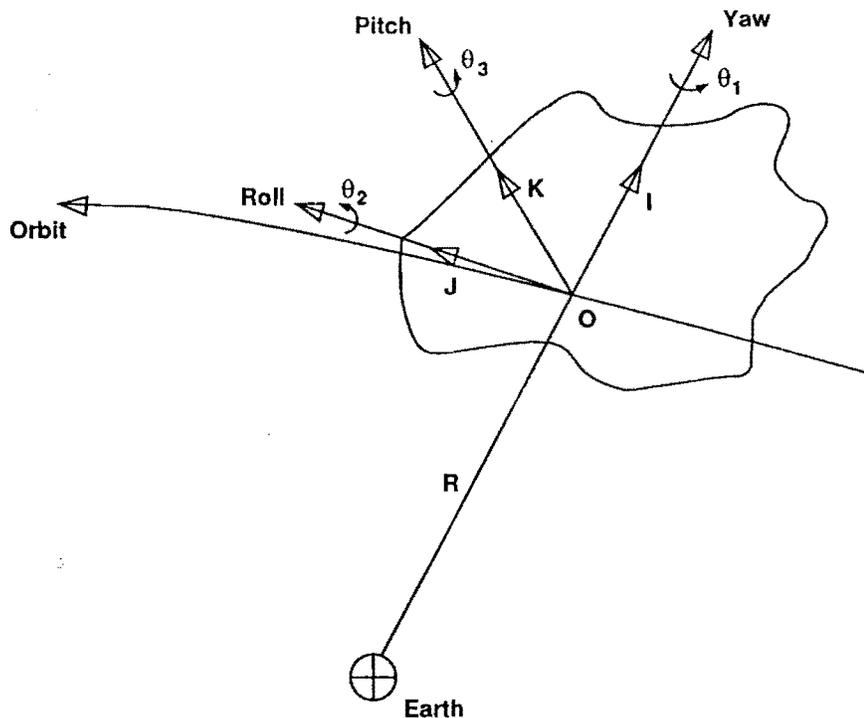
Fig. 4 Attitude Coordinate System of a Rigid Body in a Circular Orbit

Reaction wheels have only one degree of freedom with respect to the spacecraft, i.e. only the spin rate of the wheel can be changed by means of the driving motor. Reaction torques are generated about the spin axis by accelerating or decelerating a wheel from its nominal spin rate of zero. The reaction torque acts upon the stator of the drive motor, which is fixed to the satellite structure, causing the satellite to rotate in the opposite direction.

Although the direction of the control torque is fixed, motions about the principal axes are coupled due to the gyroscopic stiffness of the reaction wheel. Reaction wheel assemblies typically contain a two-phase ac servomotor designed to exhibit a relatively constant torque versus speed curve. A tachometer is included to measure wheel speed.

A reaction wheel is well-suited for cyclic torque absorption and for low torque momentum transfer during reorientation maneuvers. For these reasons, reaction wheels are especially advantageous for satellites that require a variable, though well-defined, attitude, such as astronomical or earth observation satellites.[8]

Momentum wheels are actuators that operate at a biased momentum and are used primarily to provide gyroscopic stiffness against periodic disturbance torques. A momentum wheel also

provides variable momentum storage capability about its spin axis. The capacity of a typical momentum wheel varies from 0.4 to 40.0 kg-m^2/s. Small, high-speed wheels are usually preferred due to their low size and weight.

Scanwheels have the advantage of fewer bearings, lower weight, lower power, and lower cost in their horizon sensor/momentum wheel combination.[8] These two ACS components are typically required for Earth-pointing, three-axis attitude control. The system, shown in Fig. 5, is well-suited to low-cost operational satellites that have no significant internal momentum disturbances. Scanwheel capabilities include:

- Automatic acquisition that does not require horizon sensor data, regardless of initial spin rates or orientation

- Effective, nutation damping, both before and after initial acquisition

- Automatic pitch, roll, and yaw control

- Automatic momentum control

- Accommodation of orbit-adjust disturbances without expendables
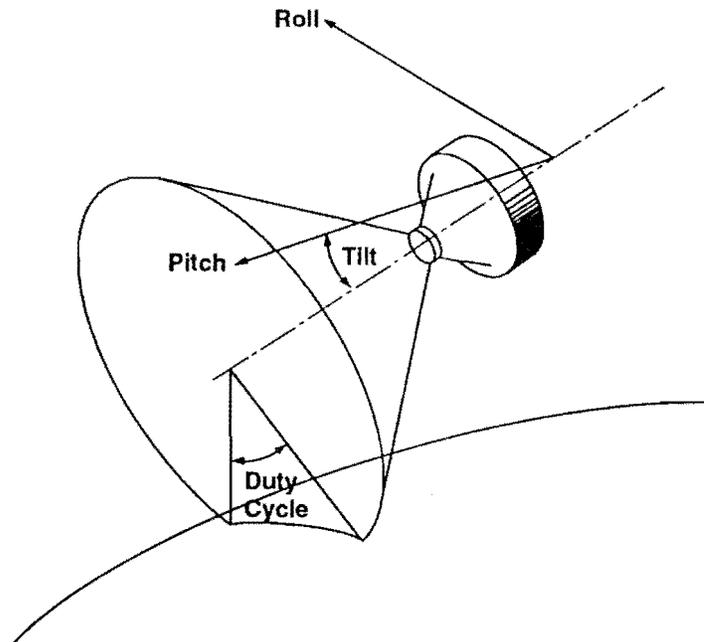
Fig. 5 Scanwheel On-orbit Geometry

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Control Moment Gyros (CMG) are either single- or double-gimballed fixed momentum devices. The single-gimbal CMG has one degree of freedom and, thus, permits control about a single axis. Three of these CMGs are required for complete three-axis control. Single-gimbal CMGs are primarily used for high-torque maneuvering or to counteract high reaction torques. Reaction torques associated with small satellites are relatively small, and vehicle moments of inertia are quite small as well. Therefore, the only benefit of a single CMG would be to provide extremely high vehicle rates during slew maneuvers. For this reason, the high performance capabilities of a single-gimbal CMG will not usually be required by current small satellites.

Double-gimbal CMGs do not have the torquing capabilities that single-gimbal CMGs provide since the outer gimbal must support the torque generated by the inner gimbal. These devices are primarily used to absorb cyclic disturbance torques.

Gimballed momentum wheels exchange momentum with the satellite by changing both momentum magnitude and direction. Cross-coupling is reduced by allowing only small changes of the nominal momentum vector. The double-gimballed momentum wheel permits a continuous rotation of the satellite if the nominal direction of the momentum vector and the satellite's rotation axis are aligned. The system also permits orientation of the satellite attitude with respect to the wheel. The three-axis control provided by the gimballed momentum wheel system is well-suited for satellites that must be aligned periodically to different ground stations.

MOMENTUM-BASED ACS CONFIGURATIONS

Various momentum-based ACS configurations have been used for three-axis stabilized satellites. Control system actuators typically consist of a combination of momentum and/or reaction wheels, thrusters, and magnetic control components. Thrusters and magnetic coils are typically used in both desaturation and active control capacities, depending on the exact configuration. Some common three-axis systems are discussed below.

<u>Three-Axis Stabilization Configurations</u>

Three-axis stabilization configuration systems are:

- Single momentum wheel
- Pitch momentum wheel/yaw reaction wheel
- Pitch momentum wheel/thruster
- Double-gimbal momentum wheel
- Single-gimbal momentum wheel
- Three reaction wheels
- Canted scan wheel momentum bias

Single Momentum Wheel System - A single momentum wheel can provide passive three-axis stabilization of a satellite with the two axes in the orbit plane being held in their position by the gyroscopic effect of the momentum wheel. Active attitude control about the third axis, which is orthogonal to the orbit plane, is obtained by increasing or decreasing the momentum wheel speed through torquing. This is normally performed in a range of ±10 percent of nominal speed or momentum bias. A damper must also be included in this system.

Pitch Momentum Wheel/Thruster System - Active, three-axis attitude control can also be achieved using a single momentum wheel mounted along the pitch axis and thrusters for pitch and combined roll/yaw control. This system is illustrated in Fig. 6(a). The coordinate axis frame for this system (and all other illustrated systems) is shown in Fig. 6(b). As a torque acts on the satellite, the wheel spin rate varies to maintain a constant attitude. If the torque is a cyclic disturbance, the wheel speed remains constant over the cycle. Secular torques acting on the satellite cause the momentum wheel speed to either increase or decrease monotonically towards saturation. The pitch control thrusters are then used to desaturate the momentum wheel by reducing its speed to the nominal operating value.

This system uses roll error signals to fire gas valves to supply control torques simultaneously about the roll and yaw axes.[9] Roll error signals are processed by a lead controller, which is best implemented by a pseudorate circuit. Restraint about the yaw axis is provided by the momentum wheel, with its angular momentum vector aligned along the negative pitch axis.

Nutation caused by the coupled roll/yaw dynamics is damped in the roll channel by the pseudorate controller. System damping is provided in yaw by offsetting the roll valves such that they supply control torques about the yaw axis. The unique feature of the one-wheel system is the use of both the offset roll-actuated control torque and the momentum wheel to control the yaw axis without a direct yaw sensor.

Single-Gimbal Momentum Wheel System - This system[9] uses a controller to shape the roll error signal and drive the gimbal angle. The gimbal axis is aligned along the roll axis, and its null position is such that the spin axis of the momentum wheel is aligned along the negative pitch axis. Rotating the gimbal from null produces a component of angular momentum along the z-axis. This system is shown in Fig. 6(c). A desaturation mechanism, such as magnetic torquers or a mass expulsion is needed. The control signal used to drive the desaturation system is typically derived from the roll angle.

Pitch Momentum Wheel/Yaw Reaction Wheel System - An alternative to the single-gimbal momentum wheel system is to use a two-wheel system, as shown in Fig. 6(d). This system has a large momentum wheel along the pitch axis (to provide gyroscopic stiffness) and a small reaction wheel aligned along the yaw or z-axis.[9]

The roll error signal is used to vary the speed of the yaw wheel. For this system, damping is directly dependent upon the desaturation gain, the magnitude of which is limited by weight and/or power constraints. Additional damping can be provided by including a yaw reaction wheel electromagnet torquer.

Double-Gimbal Momentum Wheel System - Fig. 6(e) shows the double-gimballed, one-wheel system.[9] The basic components of this system are a roll sensor signal, a compensation network, a momentum wheel mounted in two-degree-of-freedom gimbals, and a decoupling computer. Roll and yaw control torques are provided to drive the roll and yaw gimbals.
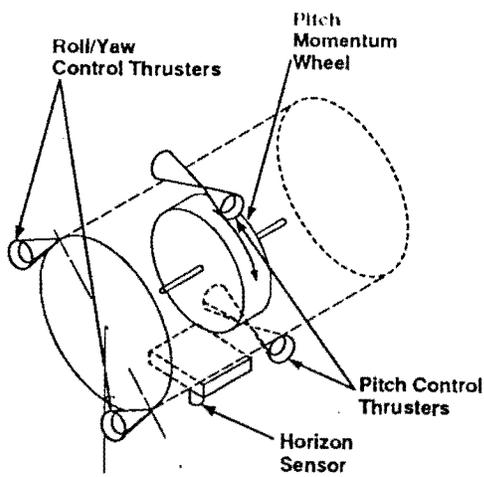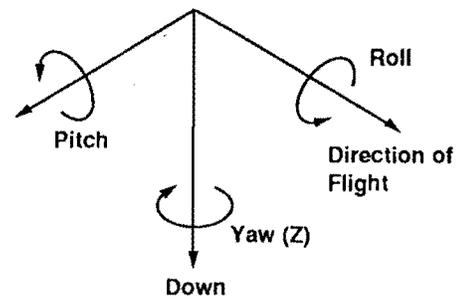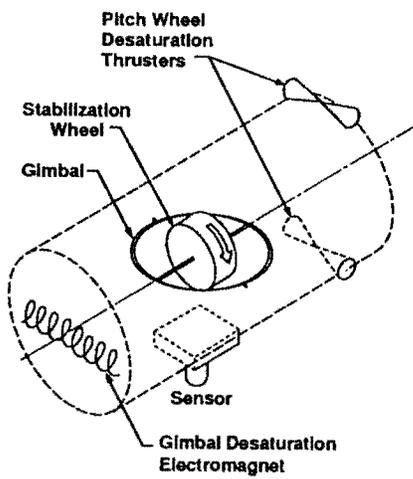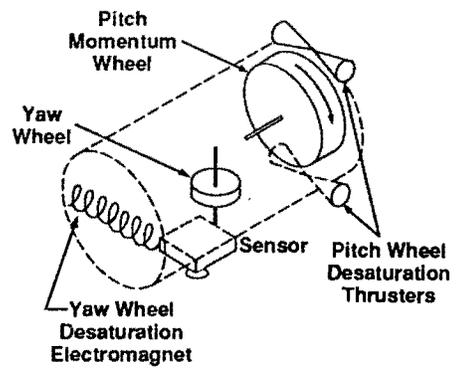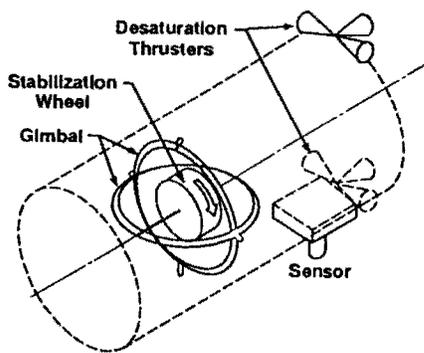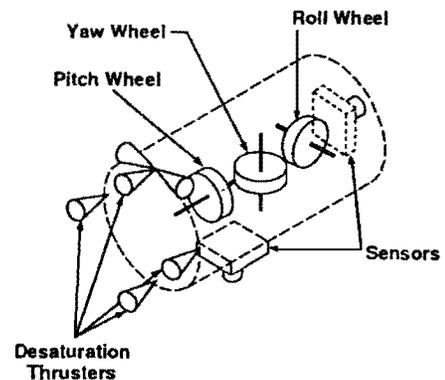
Fig. 6 Various Momentum-based ACS

The dynamic behavior of this system is similar to the one-wheel system with thrusters, except that it incorporates nutation and orbit rate decoupling. The nutation motion is decoupled in roll by subtracting the roll error from the roll gimbal drive signal. Physically, this allows the vehicle to roll while the momentum wheel's orientation remains fixed in inertial space; this significantly reduces yaw transients due to roll transients.

Roll attitude is controlled by applying a reaction torque to the roll gimbal in response to an error from the attitude sensor. The roll rate and attitude gains are determined by the parameter values in the compensation network. Yaw is controlled by a gyro-compassing technique similar to that of the one-wheel system with thrusters. As in that system, the yaw attitude is determined by the amount of gyroscopic coupling, i.e. the product $h\omega$. The yaw rate gain is produced by offsetting the roll thrusters into yaw by an angle, α. In this system, the yaw rate gain is produced by using a fraction of the roll gimbal command signal to drive the yaw gimbal.

This system can also be driven by a signal from a yaw sensor, which decouples roll and yaw motion. The yaw offset is then no longer a function of gyroscopic stiffness provided by the momentum wheel. The roll and yaw gimbals could be driven by control laws of the same type as those for the three-axis reaction wheel system described below.

Three Reaction Wheel System - Three-axis control requires three reaction wheels, since each produces a reaction torque in a single direction.[9] In its most basic form, the system, as shown in Fig. 6(f), can be considered as three parallel pitch, roll, and yaw control systems. Each system independently controls an axis by varying the speed of the reaction wheel in response to the attitude error about that axis.

Each compensated attitude signal drives a torque motor to vary the speed of the reaction wheel. The controller compensation consists of an integral term in addition to the conventional proportional and, possibly, rate terms. The integral term is an accumulation of attitude errors and serves to minimize spacecraft offset associated with external torques or internal momentum transfer. The roll and yaw channels are coupled through vehicle dynamics and there is, consequently, a continuous transfer of momentum between the roll and yaw wheels.

As the wheels absorb disturbance torques, the angular momentum changes slowly with time while the attitude remains fixed in inertia space. When the wheel reaches saturation, the angular momentum is adjusted by using gas jets or magnetic torquers. The three-axis reaction wheel system is also well-suited for carrying out slew or attitude reorientation maneuvers about a commanded axis (usually one of the wheel axes).

Advantages of a three-axis stabilized reaction wheel system are:

- capability of continuous high-accuracy pointing control

- large-angle slewing maneuvers without fuel consumption

- compensation for cyclic torques without fuel consumption

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Canted Scanwheel Momentum Bias System - An alternative design[7] for a momentum bias control system has a pair of canted scanwheels in the pitch-yaw plane, as shown in Fig. 7. The scanwheels use the pitch and roll attitude error signals to maintain closed-loop, three-axis attitude control. The pitch and yaw momentum conponents are given by:

$$h_p = (h_1 + h_2) \cos \alpha \tag{10}$$

$$h_y = (h_1 - h_2) \sin \alpha$$

where α is the cant angle between the pitch axis and the momentum wheels. The momentum wheels are nominally operated at the same speed, such that $h_1 = h_2$, and the total momentum is along the pitch axis, with $h_y = 0$. When the horizon scanners sense a roll angle error, a controlled yaw momentum component is generated by differentially torquing the two wheels to reduce the anticipated yaw error, which will occur one-fourth of an orbit later.
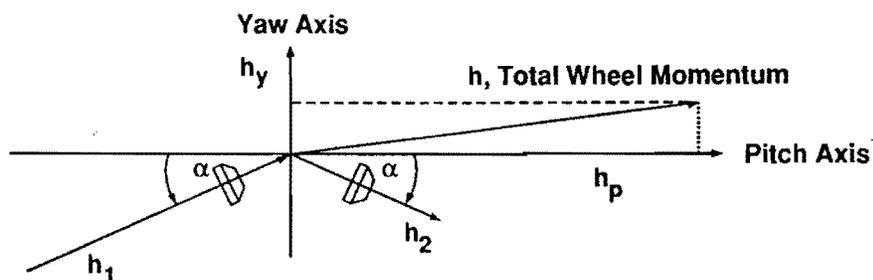
Fig. 7 Canted Momentum Wheels in the Pitch-Yaw Plane (h_1 and h_2 are the wheel momenta)

CURRENT SMALL SATELLITE ACS CONFIGURATIONS

The following is a brief description of three-axis stabilized ACS configurations that use momentum concepts.

SMEX (Small Explorer or Scout Explorer)[4]

Sun, star, and vertical pointing are provided by a momentum stabilized control system consisting of two orthogonal momentum wheels and various sensor suites. For vertical pointing, conventional horizon scanning and gyro compassing provide a local vertical orientation of one degree or better and yaw to within two degrees.

Star pointing requires the addition of a star tracker and a gyro package, each with two axes of error sensing. The two axes are aligned normal to the instrument pointing axis. The momentum

vector is pointed toward the sun, not toward the target, to permit rapid reorientation of the instrument. The two wheels are mounted in the plane of the instrument pointing axis and the sun line. The sun-target angle is varied by adjusting the ratio of the wheel speeds, while the transverse axis is controlled by altering the total wheel momentum. Thus, the two axes of errors are transposed to (roughly) the sum and difference of the wheel speeds. Sun pointing is a subset of star pointing in which the star tracker is replaced with a sun tracker.

BSS (Ball Scout Satellite)[1]

Attitude sensing, control, and determination are handled through a flexible set of components. The basic satellite control accuracy is to $\leq$0.2 deg, with knowledge to $\leq$0.1 deg. For precision pointing missions (e.g., star staring, planet/comet tracking), the precision payload is used to pass sensed pointing errors to the BSS in lieu of expensive precision trackers. If the payload provides error signals to arcsecs, the BSS can control the attitude to arcsecs.

BSS pointing missions use magnetic torquing and reaction wheel(s). A single scanwheel is used for nadir and zenith pointers. Polar nonsync nadir/zenith orbiters require adding a yaw wheel. Elliptical orbits require the addition of another horizon scanner. Optional sun sensors provide improved ground determination. The celestial three-axis controlled pointers use sun sensors and a single pitch wheel for slow retargetting and two wheels for fast retargetting.

MAGSAT (Magnetic Field Satellite—APL)[10]

The ACS maintains orientation to the local vertical coordinate system by using a momentum-biased infrared scanner/reaction wheel (scanwheel) and magnetic torquing for initial orientation and occasional control. A pitch axis rate gyro, sun sensors, coarse magnetometers, and the horizon scanner provide attitude and rate information ground or on-board semiautonomous microprocessor control. A three-axis set of coils provides magnetic torquing, but operation of the coils is absolutely minimized during the mission. A nutation damper and in-flight adjustable aerodynamic trim surface (used to balance aerodynamic torques) complete the ACS elements.

WHEEL SELECTION AND SIZING

Wheel Selection Model[9,11]

Four missions have been selected for analysis of momentum-based attitude control systems for small satellites. These missions are defined in Table 1. Note that the choice of moments of inertia for Missions A and B was based on the assumption that large solar panels were required. The satellite design for Missions C and D does not include solar panels, and mass moments of inertia are equal.

Table 1

MISSIONS FOR WHEEL SELECTION MODEL

Parameters	Mission A	Mission B	Missions C and D
Mission type	Telecommunication	Telecommunication	Meteorology
Accuracy	All axes 0.07 deg	Pitch and roll axes 0.05 deg, yaw axis 0.5 deg	All axes 0.07 deg
Orbit	Geostationary	Geostationary	1000 km circular, polar
Mass	500 kg	500 kg	500 kg
Moments of inertia	Pitch 300 kgm², roll, yaw 1500 kgm²	Pitch 300 kgm², roll, yaw 1500 kgm²	All axes 300 kgm²
Mission life	15 years	15 years	3 years

Table 2 lists momentum and torque requirements resulting from mission-specific disturbance and reaction torques for each of the four cases.

Nominal wheel momentum is determined by both required momentum storage capacity and required yaw accuracy. The sinusoidal component of the momentum storage capacity is determined by:

$$h_{STORAGE} = \frac{a}{\omega_o}$$

$$\omega_o = \sqrt{\mu/R^3}$$

(11)

where a = maximum amplitude of the sinusoidal torque

μ = gravitational constant

R = radius of a circular orbit

ω_o = orbital rate

Table 3 presents typical values of a and ω_o for different missions.

The momentum storage capacity must also be sufficient to store both secular and short-duration torques. Therefore, a momentum storage capacity between 1 and 2 N·m-s is required. This capacity can be reduced by optimizing the desaturation system.

For a gimballed system, rotor speed will normally be varied by about 10 percent, and gimbal deflection will be less than 6 deg during the periods that require high accuracy. This means that the nominal wheel momentum must be ten times greater than the required storage capacity.

Table 2

MOMENTUM AND TORQUE REQUIREMENTS

Requirement	Mission A, B		Mission C, D	
	Torque (N·m)	Momentum (N·m)	Torque (N·m)	Momentum (N·m)
Gravity Gradient	2.5×10^{-8}		8×10^{-8}	
Solar Radiation	4.6×10^{-6}		9.2×10^{-7}	
Antenna Radiation	5×10^{-7}		5×10^{-8}	
Aerodynamic	Negligible		10^{-6}	
Magnetic	Negligible		Negligible	
Meteoritic		10^{-2}		1.5×10^{-5}
Solar Paddles	0.03	0.1	—	—
Other Moving Equipment	0.01	0.1	0.01	0.1
Station-keeping Jets	0.1	2.5	—	—
Station-keeping Jets (Electrical)	2.5×10^{-4}	2.5	—	—
Desaturation Jets	0.1	<1	0.1	<1
Desaturation Jets (Electrical)	0.01	<1	—	—

Table 3

TYPICAL VALUES FOR DIFFERENT MISSIONS

Mission	Torque Amplitude a (N·m)	Orbital Rate ω_0 (rad/s)	Storage Capacity a/ω_0 (N·m)
A, B	1.4×10^{-5}	7.29×10^{-5}	0.2
C, D	4.5×10^{-6}	1.00×10^{-3}	4.5×10^{-3}

For a system with one wheel mounted along the pitch axis, the steady-state yaw offset error (ψ_{ss}) is given by:

$$\psi_{ss} = \frac{T_z - T_x \tan \alpha}{\omega_o h} \tag{12}$$

where T_x and T_z are disturbing torques about roll and yaw axes, respectively; α is the offset angle of roll-yaw coupled control thruster; and h is the angular momentum of the pitch wheel. The angle α is typically only a few degrees. Therefore, the roll torque contribution is much less than the effects of the yaw torque and can be neglected for preliminary design studies. Thus, the peak yaw angle due to an external torque is given approximately by:

$$\psi \approx \frac{T_z}{\omega_o h} \tag{13}$$

This equation can be used to approximate required wheel momentum. Rearranging this expression and solving for wheel momentum gives:

$$h = \frac{T_z}{\omega_o \psi} \tag{14}$$

For a desired yaw accuracy, this equation can be used to size the angular momentum of the pitch wheel. Selecting the maximum continuous torques for Missions A and B (solar pressure) and C and D (aerodynamics), the angular momentum can be plotted versus yaw accuracy. The results are presented in Figs. 8 and 9.

For geosynchronous Missions A and B (Fig. 8), an increase in wheel momentum beyond 9 N·m-s does not result in better yaw accuracy. For LEO Missions C and D (Fig. 9), the angular momentum ranges from 1 to 2 N·m-s, and a large gain in yaw accuracy can be achieved by a relatively small increase in wheel momentum.

A rule of thumb for the relationship of the weight of the wheel housing and associated electronics to the angular momentum for both reaction and momentum wheels is

$$W = 3.2h^{0.4} \tag{15}$$

where W is in kg and h is in N·m-s. Note that this equation can be used for all four missions and agrees well with published data.[9]

After sizing the angular momentum, Eq. (15) can be used to approximate the weight of the wheel system. The result is depicted in Fig. 10, which shows that to obtain better yaw accuracy, a heavier momentum wheel must be selected. Also, for the same yaw accuracy, the higher orbit altitude requires a heavier momentum wheel.
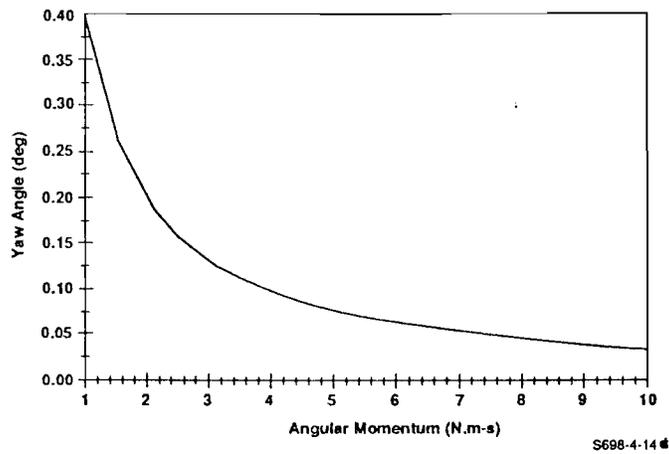
Fig. 8 Yaw Pointing Accuracy for Missions A and B
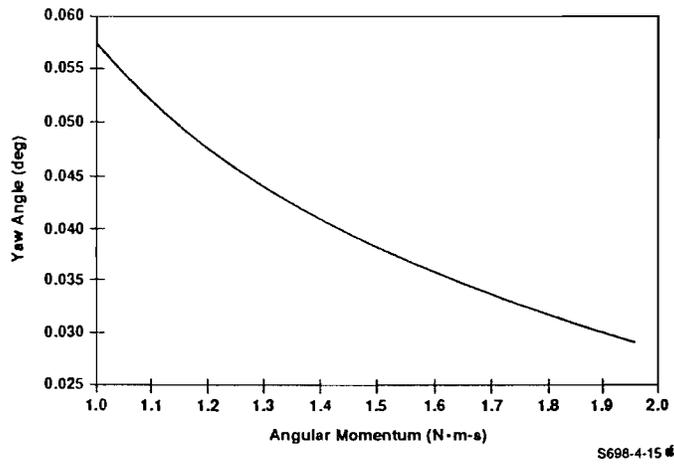
Fig. 9 Yaw Pointing Accuracy for Missions C and D
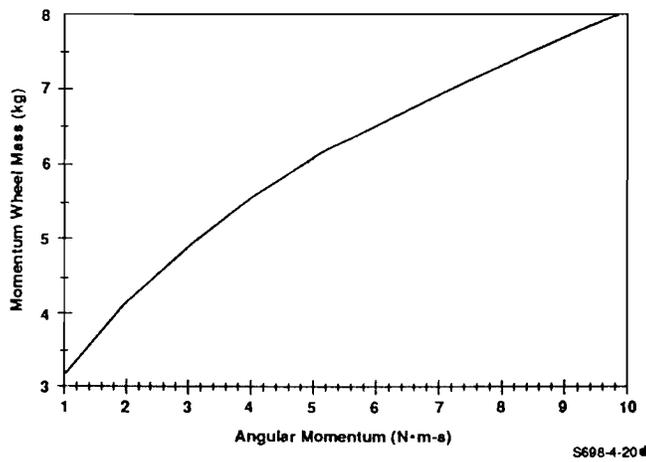
Fig. 10 Wheel System Mass As a Function of Angular Momentum

DESATURATION TECHNIQUES

Secular torques acting on a vehicle cause the wheel speed to increase until it reaches its upper operating or saturation limit. At this point, an external torque must be used to restore the wheel speed to its nominal operating value so that the wheel is maintained in its torque-producing range. The two most common techniques used for desaturation (magnetic torquing and mass expulsion torquing) are described below.

Magnetic Torquing

Electromagnetic actuation systems are quite common and have been discussed and analyzed in several papers and reports.[12] Three types of magnetic torquer systems, currently being used, are permanent magnets, air-core torquing coils, and iron-core torquing coils. However, due to the weight of permanent magnets, their use is limited. In most systems, a set of three mutually perpendicular coils is used. The control torque $\underline{T}$ is given by $\underline{T} = \underline{d} \times \underline{H}$, where $\underline{d}$ is the magnetic dipole moment, and $\underline{H}$ is the magnetic field strength or magnetic intensity.[7]

In a magnetic desaturation system, momentum dumping is exercised when the wheel speed exceeds a specified threshold. The commands are generated either manually from a ground station or automatically by an on-board computer. The magnetic coils are energized at predetermined orbit locations and remain on until wheel speed returns to its nominal value. Magnetic desaturation systems also require a three-axis magnetometer and signal processing electronics. The magnetometer sensors detect Earth's magnetic field components along each of the satellite axes. The corresponding signals are amplified and used to drive the coils.[13]

The magnetic coil system is an attractive method of desaturation for low earth orbit (LEO) satellites due to the relatively high magnetic field intensity at lower elevations. This system also has high reliability since it includes only simple static devices (a magnetometer, a signal processor, and three coils). Other advantages are that it does not depend on a fuel supply and is much lighter than the simplest low specific impulse thruster system. Disadvantages of this system are that it may require significant amounts of power at higher altitudes. Also, coil commands may last over a large fraction of the orbit (or over several orbits) to reach desaturation. Magnetic systems may also interfere with the operation of certain payloads.

Coil systems are also used to provide libration damping for a gravity-gradient satellite. An example of this application is the Globesat GS-100 satellite, a two-axis gravity-gradient stabilized satellite in a 500-km circular orbit.[14] The GS-100 satellite is equipped with three magnetic torquers to generate rotational impulses. Fig. 11 shows the location of the coils, which are mounted inside the satellite. The coils are made of 24 AWG copper; the yaw coil has 100 turns and weighs 0.78 kg; the roll and pitch coils each have 80 turns and weigh 0.93 kg. The maximum torque capability of each coil set is approximately 1.4×10^{-3} N·m in a 0.3-gauss field. The power consumption is about 36 watts per coil set and about 3 amperes at 12 volts.
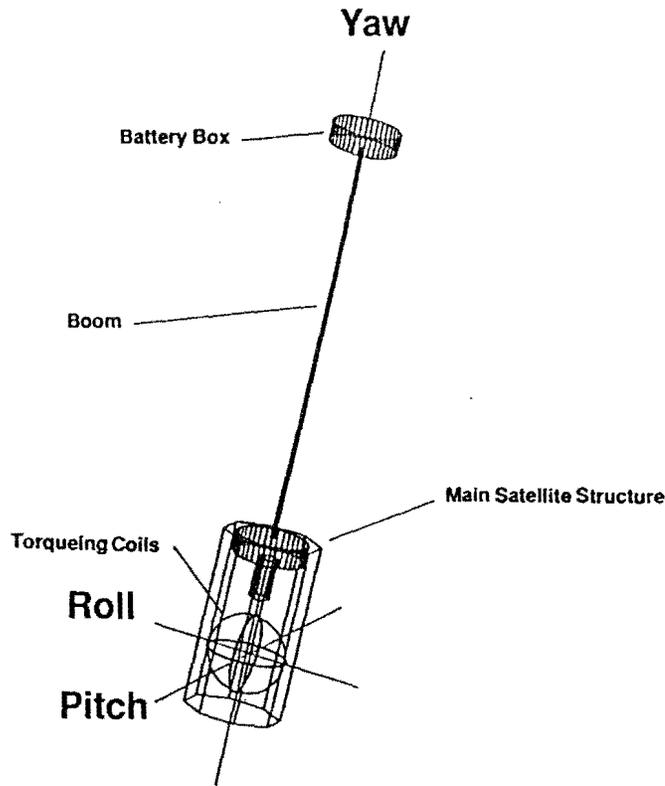
Fig. 11 GS-100 Satellite with Magnetic Torquers

Mass Expulsion

Mass expulsion systems are used for desaturation of momentum storage systems. In operation, a jet is fired to produce a torque opposite to the direction of the accumulated angular momentum while the satellite is commanded to maintain its attitude. This results in a wheel acceleration that counteracts the applied torque.

The desired desaturation torque $\mathbf{T}_D$ is produced by applying the appropriate thrust force ($\mathbf{F}$) at its fixed momentum arm ($\mathbf{d}$). The resulting torque, applied about the satellite's center of mass, is given by

$$\mathbf{T}_D = \mathbf{d} \times \mathbf{F} \tag{16}$$

This equation contains:

- magnitude and direction of the applied torque

- length of the momentum arm

- time between each firing

- firing period

These parameters can be sized and calculated easily when the desired pointing accuracy, satellite moments of inertia, and orbital elements are available.

For a desaturation system using body-fixed offset roll/yaw thrusters, the efficiency of the system can be defined as the ratio of the daily secular momentum increase to the angular impulse provided by the thruster. A reasonable design value for the efficiency is about 80 percent.[9]

The duration of the desaturation impulse is a function of the amount of momentum to be dumped. This is typically about one percent of nominal wheel momentum.[11] For a 500-kg satellite in geosynchronous orbit, the required torque for desaturation is about 0.01 N·m-s.[8] The number of thruster cycles expected over the lifetime of the satellite is a potentially limiting item. Another concern is that a wheel system will tend to exhibit a nutation when within the thruster deadbands. The amplitude of this nutation is dependent on the minimum impulse bit from the thruster. Reducing the impulse bit to decrease the nutation amplitude will increase thruster cycles.

Thruster systems typically include a number of solenoid valves, nozzles, high-pressure tanks and lines, as well as fuel heaters and pressure transducers. The complexity of these systems reduces reliability and increases weight. Thrusters for momentum desaturation also need to be rated at a few millipounds of thrust to avoid significant disturbances to the wheel system. In high altitude orbits, however, mass expulsion systems are the only viable means for desaturation due to the low magnetic field intensity at higher elevations.[15]

ACS IMPROVEMENTS AND COST REDUCTIONS FOR SMALL SATELLITES

The attitude control system technologies discussed in this paper were initially developed for expensive satellites. Since target costs for small satellites are between one and ten million dollars, the cost of ACS components must be consistent with this overall budget.

The majority of ACS components is subjected to rigorous and costly testing and to reliability, lifetime, and traceability requirements that are endemic to the aerospace industry. The costs associated with these requirements are reflected in component costs and, therefore, prevent widespread use of many flight-proven ACS components in small satellite applications. Since small satellites typically have less-rigorous requirements, a significant cost reduction in ACS components could be realized through reducing component processing while maintaining sufficient quality control.

Most ACS components have associated electronics that are typically required to be S-level. The costs of many S-level components make them unfeasible for small satellite applications. In addition, applications of new technologies (and even not-so-new technologies) have not yet been realized due to the exorbitant costs associated with qualifying a part for use in space.

The electronics associated with many of the momentum-based ACS components discussed are typically comprised of discrete components. The functions of these components could be combined into a standard integrated circuit (IC). The development cost of a custom IC may be substantial, and aerospace customers are hesitant to fund such an endeavor. However, long-term benefits for both expensive and cheap satellites could be realized through reduction in electronics assembly time and elimination of assembly error.

Another enticing prospect is the utilization of radiation-hardened, RIC-MOS technology currently being developed at Honeywell's Solid-State Electronics Division. A RIC-MOS chip could be easily developed to be interchangeable with a commercial-level IC. With standard interfaces, a portion of the small satellite's electronics could become radiation-hardened through a simple IC chip interchange.

With widespread use of microprocessor-based attitude control systems comes the need for standardization and flexibility of bus architectures. Microprocessor tasks in a reaction wheel assembly are:

- I/O control

- Telemetry monitoring

- Motor commutation

- Close interval loop (torque loop)

Possible standard bus candidates include IEEE 488, 1553, or even RS 232. In most applications, the processor is standard architecture, and the PROM can be changed easily to fit the specific application. Space transportation architecture studies have shown that costs over lifetime can be reduced by standardizing interfaces and package sizes in a way similar to what the avionics industry did years ago. As parts evolve and new parts come on-line, dedicated wiring interfaces are the first to need redesign. Redesign, however, can be minimized through flexible architecture.

CONCLUSIONS

The selection of a small satellite attitude control system and its components is not a simple task. Many factors must be considered when choosing an ACS configuration for a given mission. This paper has presented an overview of these factors, with emphasis on momentum and torque requirements.

A detailed discussion of available momentum devices for use in small satellite attitude control systems has shown them to be advantageous for the following reasons:

- Concepts are flight-proven.
- Operation is independent of orbit and environment.
- They permit high pointing accuracies.
- They reduce (or eliminate) expendables.
- Components can be mixed and matched to suit specific requirements.
- Components have relatively low mass, power, and volume requirements.

Momentum-based attitude control systems have been used on various small satellites. Widespread use, however, will depend on an assessment of user specifications and subsequent relaxation of stringent and costly requirements. Additional cost savings can be realized through standard interfaces and designs built around off-the-shelf components.

Utilizing commercial electrical components will also reduce costs. The reliability associated with these components is consistent with lifetime expectancies and performance requirements associated with most small satellites. Experimental, low-cost small satellites can be used to flight test state-of-the-art electronics and new attitude, determination, and control concepts.

There is a certain amount of assumed risk that is associated with the cost-saving measures presented here. However, as these cost savings are realized, small satellite production volume will increase, and the level of risk will be justified.

ACKNOWLEDGEMENTS

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